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# THE SOLAR-POWERED SPACE SHIP

by

Krafft A. Ehricke



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# THE SOLAR - POWERED SPACE SHIP

by

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## ABSTRACT

The characteristics of different propulsion systems for space vehicles are discussed and compared. The solar-powered space ship is evaluated in greater detail. A new light-weight design is presented, using spherical reflectors. Problems of design and operation are discussed by example of a hydrogen-operated prototype design. A number of basic characteristics of the vehicle system is established. A theor tical analysis of the spherical reflector as energy collector is presented.

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## Nomenclature

| | |
|---|---|
| A | Area |
| F | thrust |
| g | gravitational acceleration |
| $\Delta h$ | enthalpy difference |
| $I_{sp}$ | Specific impulse (lb thrust sec/lb fluid) |
| $Q$ | Total radiation energy flux density of reflector. |
| $q$ | flux density for unit minimum area (cf. appendix) |
| R | reflectivity at normal incidence |
| R | reflectivity |
| r | radius of the spherical mirror |
| S | solar constant ( $\frac{2}{60}$ cal/cm$^2$ sec) |
| T | temperature |
| $\sigma$ | Stefan Boltzmann constant<br>(0.56686 $10^{-4}$ erg/cm$^2$ deg$^4$ sec<br>$1.354 \cdot 10^{-12}$ cal (mean)/cm$^2$ °K sec) |

For additional notations cf. Fig. 11.

## 1. Introduction

The enormous cost of supplying spaceborne vehicle systems from the earth is one of the principal restrictions in space flight with chemically powered propulsion systems. The versatility and freedom of operations in space will increase to the extent to which the dependency on terrestrial supply of propulsion material can be reduced.

The dependency on terrestrial supply has two aspects, the first pertaining to the energy source, the second to the expendable matter. Thrust is produced by energising matter, theoretically even to the point of converting it into radiation (1)[2]. In all other, less extreme cases, thrust is produced by accelerating expendable matter. As the exhaust velocity increases, the mass consumption decreases for a given operation; but the energy consumption of course increases. Thus, one faces the engineering problem of reducing the mass consumption (hence the terrestrial supply requirements) and increasing the energy supply and still maintaining reasonable overall operating conditions. This problem is not solved by simply pointing out another possibility for producing enormous exhaust velocities. One must also ask under what conditions of energy supply and thrust per unit weight these exhaust velocities can be obtained. In many instances a more detailed analysis shows that the attainment of such exhaust velocities is the least of the engineering problems involved and that the energy source, energy conversion etc. are now much bigger headaches; in other words, the problems are just shifted into another area, because one does not obtain anything for nothing. If one deals with such questions for a while, once comes to feel somewhat apologetic towards the "good old" chemical rocket which in many respects is indeed hard to beat.

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[2] Numbers in parenthesis refer to References on page

This is not intended to be a criticism of attempts to investigate other propulsion systems, but rather to caution lest one loses a realistic perspective towards the facts of space propulsion or feels that further hard work in improving the chemical system can be dispensed with in favor of more advanced systems. Space flight will become a reality through the chemical propulsion system. On the other hand, however, it is a fact that the supply requirements of the chemical system are a real handicap to anything more than occasional space expeditions. This shortcoming is significant enough to encourage research and analysis of alternate propulsion methods. The present paper attempts to make a contribution to these efforts.

## 2. Propulsion System Characteristics

With the possible exception of the hypothetical total photon propulsion, the main difference between chemical propulsion and all other methods of propulsion is the separation of energy source and working fluid, hereafter briefly called medium (Fig. 1). In the chemical system this consolidation of energy and medium provides smooth (comparatively) and rapid conversion, but limits the selection of media and maximizes the dependency upon terrestrial supply, inasmuch as oxidizers and fuels are nowhere readily available under natural conditions.

Separation of energy source and medium results in more flexible systems, but also raises new problems regarding energy supply, energy conversion and equipment weight. Fig. 1 presents a number of these systems which already have been suggested before on several occasions, e.g. (2) to (9), with the exception of the arc heating system which however has been studied previously for the purpose of producing an ultra-high-speed gas flow in test facilities (10,11).

Briefly, the principal potential pros and cons of separation of energy source and medium can be summarized as follows:

Advantages:

(a) Possibility of higher energy concentration than in chemical propellants through the use of nuclear power sources (e.g. pile, $\beta$-decay battery).

(b) Possibility of permanent energy supply from the sun, at least in terrestrial and intra-terrestrial space. The energy supply, however, is much less concentrated then in either nuclear or chemical sources.

(c) Greater freedom in the selection of media.

(d) Possible simplifications of the propulsion system where only one type of fluid is used.

(e) Possibility of supply of medium from other sources than the earth (e.g. refilling of the Saturn moon Titan or on Jupiter moons), thereby increasing the range of the ship for a given terrestrial supply.

Disadvantages:

(f) Greater complexity and weight of the energy source, energy conversion and transfer mechanism and related equipment.

(g) In many cases very low thrust-to-weight ratios, hence the danger of high gravitational losses when operating near planets as well as poor manouverability. Application restricted to space only.

(h) Difficulties in handling and maintenance of a nuclear energy source (pile).

(i) Operational difficulties resulting from excessive energy release necessitated by low conversion efficiencies, such as the need to dispense with excess heat in conversion systems.

The last mentioned disadvantage becomes the more severe the higher the ultimate energy level prior to conversion is supposed to be, that is, the more energy is available prior to expansion and/or discharge of the medium. Figure 2 presents a survey of the energy converted in producing thrust versus the specific impulse for various propulsion systems. For reasons of comparison the equivalent energy for different flight mechanical energy levels is given. The relatively flat slope of the conversion line shows that any increase in specific impulse must be bought at considerable increase in energy imparted to the jet (in fact, $I_{sp} \propto \sqrt{\Delta h}$ ). This is the reason for the problem shift towards the energy source and conversion system as the specific impulse goes up; and obviously, if 10 kcal/g instead of 1 kcal/g is required at 0.25 overall conversion efficiency, the production of 40 kcal/g at the source and the need to dispense with 30 kcal/g is much more of a problem than the production of 5 kcal/g and the need to remove 3 kcal/g. Thus, high specific impulse is desirable, but it must be in proportion to the flight mechanical energy requirements (i.e. not greater than necessary and convenient for the contemplated mission).

A survey of some basic characteristics of these propulsion systems is presented in Tab.1. Most items are self-explanatory. It should be pointed out that the specific energy consumption in kw per lb thrust was evaluated as follows:

The production in the propulsion system to get the thrust producing mechanism started, the energy converted from its original form into thermal energy, and finally the energy converted in the jet proper, which is equal to $\Delta h$(Btu/sec)/0.984 $I_{sp}$ (lb sec/lb), where $\Delta h$ and $I_{sp}$ are taken from Fig. 2 and 1/0.984 is the conversion factor from Btu to Kw. The figures pertaining to the chemical system reflect the fact that little has to be produced outside the medium (only auxiliary power) while the medium itself releases the energy. The conditions are reversed in all other cases where energy must be transferred to the jet.

Against this general background the solar-powered space ship will be discussed in more detail.

## 3. The Solar Powered Space Ship

Table 1 shows that among the possible non-chemical systems, the solar-powered drive is fundamentally the simplest and most straight forward arrangement. It uses an existing energy source in space in an efficient manner, by converting the radiant energy directly into heat. The propulsion system simply consists of tankage for one type of fluid, radiation collector, heat exchanger and exhaust nozzle. Pumps are required to circulate the working fluid, but since the quantities involved are very small (less than a pound), the pumps are light, the horsepower requirement is low and electrical high-speed drives appear practical. The necessity of concentrating the thin-spread solar energy requires the use of large reflector-type collectors. In this manner very high heat flux densities can be obtained which are equal and greater than those normally found in the throat of chemical rocket engines ( $> 3$ Btu/in$^2$sec). Most fluids are heated to a very high temperature under these conditions and this makes their subsequent piping to the exhaust nozzle quite difficult. If hydrogen is selected, the high heat capacity permits to store a considerable amount of energy in the fluid, while keeping the temperature within such limits (1,300-1,500°F) as to permit the use of uncooled pipelines for the heated material, while the specific impulse nevertheless is the highest attainable with any fluid under these conditions and, in fact, exceeds that of chemical propellants (Figures 3,4). By using uncooled pipelines, the system can be simplified and its weight kept low.

Low weight is of extreme importance, since the thrust obtainable is only of the order of 100 to 200 lb. Even in the case of orbiting space ships the thrust-to-weight ratio cannot be allowed to become arbitrarily low. The lower limit is rather determined by the specific impulse available and by the strength of the gravitational field in which the system is to operate. In the case of the earth-moon field and with a specific impulse around 450 sec, the thrust-to-weight ratio should not fall appreciably below 0.01g, because otherwise the flight mechanical performance during the powered phase becomes so poor that impractically large mass

ratios are needed to perform a cislunar flight mission. This is due to the gravitational losses incurred in a slow-climbing spiral-path associated with very low thrust. However, low thrust is not only a disadvantage. It is in fact mandatory for the system under consideration, because the considerable size of the collectors yields very long moment arms. At thrust-to-weight ratios of 0.1 to 0.2 g it would not be possible, within reasonable weight limits for space vehicles of this size, to provide the necessary structurally rigidity and to prevent bending and distortion which would destroy the optical quality of the reflectors.

A compromise between these two opposite thrust requirements must be established and leads to values of the order of 0.01 g. If the right conditions are fulfilled, the solar-powered drive will require much less working fluid supply from the earth than any chemical propulsion system. The limited power of this system makes its use for interplanetary flights unlikely. It appears suitable for cislunar and lunar operations which presumably will be more frequent and for which a reduction in fluid supply is therefore quite important.

These considerations make the solar-powered sytem appear attractice. However, there are mery problems to be solved if its usefulness is to be assured. The principal design problem lies in the extreme emphasis for all-out light-weight construction. Another design as well as dynamics problem is the requirement for completely independent orientation in space of the optical axis and the thrust axis with respect to each other. The problem of rigidity and the resulting autopilot control difficulties can be reduced to practical values by means of proper bracing of the collectors. Considerable pressure losses are introduced by long pipelines between tank, heater and motor, requiring additional pumping energy. This difficulty is apparently unavoidable. A large number of different designs has been evaluated with the purpose of reducing the length of piping, before the design presented in this paper was selected. Basic shortcomings of the system are of course dependency on solar radiation and the possibility of damage even by small meteors due to large size and frail construction. The vehicle is

without means of propulsion while in the shadow of a celestial body. The effect of cosmic dust may be the most serious of all problems and may in fact constitute a hurdle which cannot be overcome. It is too early, however, to assess the effect of dust in space with sufficient certainty to make such a statement. Furthermore, the density and distribution in cislunar space must be known more accurately. It is not to be expected that the distribution will be uniform in view of the complex interaction of terrestrial, lunar and solar gravity fields in cislunar space.

Further technical problems will become apparent during the subsequent discussion. Altogether they make the solar-powered space ship difficult to realize; however, while constituting a great challenge to engineering ingenuity, the successful solution of its problems will reward us with increased freedom in space.

## 4. The Working Fluid

The selection of the working fluid is mainly determined by the requirement for low overall vehicle weight and the definite need for high specific impulse. Desirable qualifications derived from the preceding discussion are high heat capacity and low condensation temperature to permit a maximum degree of expansion and high energy conversion at relatively low initial temperature. In order to maintain a high heat transfer coefficient throughout the heating period, the working fluid must be in supercritical state. Therefore a not too high critical pressure is desired.

The working fluids which best meet these specifications are hydrogen and helium. Their relevant characteristics are summarised in Table 2. Hydrogen yields higher specific impulse, because its molecular weight is lower and its specific heat higher. Helium has the advantage of higher density and of lower critical pressure as well as higher critical density. It has also a lower heat of vaporisation. The density impulse of helium is about 10 percent higher than that of hydrogen.

However, it is believed that hydrogen is in this case the only choice which is acceptable from practical considerations. It is more readily available, it permits to operate the propulsion system at a much lower temperature level at about equal motor performance and it yields a lower gross weight, hence a higher thrust-to-weight ratio and less gravitational losses. At present, sufficient practical experience in the use and pumping of hydrogen in rocket engine systems is available to permit estimates for the technical layout of a solar-powered propulsion systems. The molar heat of hydrogen under different supercritical pressures is shown in Fig.5. These curves were obtained by measuring the slope $(dS/dT)_p$ of curves in the hydrogen entropy-temperature diagram ref. (14) and computing the molar heat capacity from $C_p = (dS/dT) \cdot T$.

An estimate of the density of liquid hydrogen under the slightly supercritical pressure of 15 atm is shown in Fig. 6. The variation of exhaust temperature $T_e$ and enthalphy $\Delta h$ converted in the nozzle is shown in Fig. 7 as function of chamber pressure and chamber temperature. Figure 7 also shows the enthalpy $\Delta h^*$ required to convert the hydrogen from its original state in the tank to the state prior to expansion in the nozzle. Because of the relatively low temperatures involved, dissociation is neblfgible and $\Delta h^*$ is not a function of the pressure prior to expansion. By dividing $(\Delta h / \Delta h^*)_T$ one obtains ratios between 0.75 and 0.85 at supercritical pressures.

## 5. The Energy Collector

In order to increase the enthalpy of the working fluid by the required amount in a short period of time, the radiation energy must be concentrated in a small area in which the heater is placed. Usually, parabolic reflectors are considered for this purpose (4, 7). In a parabolic reflector the radiation is concentrated in a small focal area where extremely high temperatures or heat transfer rates can be obtained. Farber and Davis (16) have analyzed the parabolic reflector for the purpose of attaining maximum temperatures in the focus. They find the highest theoretically possible temperature of a black body receiver to be 5,100°K (assuming 100 % reflectivity), compared to 6,000°K solar surface temperature.

In a solar propulsion system the use of a collector is not to attain highest heat transfer rates in the smallest area. The purpose of the collector is to distribute energy over a certain area so that sufficient time is given the fluid to absorb energy while at the same time the flux density remains high enough to produce a source-sink system of adequate intensity for the temperature to be attained by the fluid. In order to do this with a parabolic reflector, its optical quality must be reduced purposely.

However, the main argument against the use of a parabolic mirror comes from weight considerations in relation to the energy obtained or thrust generated per unit of area intercepted. The radiation energy collected by a reflector is equal to the area intercepted, A, times the solar constant, S, and the reflectivity at normal incidence, $R_n$,

$$Q = A\ S\ R_n \qquad (1)$$

From Fig. 7, $\Delta h^*$ for $T_o$ = 1,000°K, neglecting losses, is 3,557.5 calories per gram of hydrogen. The solar constant in space is S = 2/60 cal/sec cm$^2$. Thus, if

one assumes roughly that all energy intercepted can be transferred to the working fluid (hydrogen), one obtains for the reflector area per unit weight of hydrogen per second, heated to 1,000°K,

$$\frac{A}{(\dot{W}_{H_2})} = \frac{106,725}{R_n} \quad \frac{\text{cal sec cm}^2}{\text{g sec cal}} = \frac{\text{cm}^2}{\text{g }(H_2)}$$

$$\left.\begin{aligned} &= \frac{10,672.5}{R_n} \quad \frac{\text{m}^2}{\text{kg}(H_2)} \\ &\phantom{=} \frac{52,190}{R_n} \quad \frac{\text{ft}^2}{\text{lb}(H_2)} \end{aligned}\right\} \tag{2}$$

From Figure 3 one obtains for a pressure ratio of 15/0.1 atm through the nozzle and $T_c$ = 1,000°K, a specific impulse of 478 sec. . Assuming 94 percent or 450 sec, one obtains for the reflector specific impulse

$$\frac{A}{I_{sp}} = \frac{116}{R_n} \quad \frac{\text{ft}^2}{\text{lb(thrust)}} = \frac{23.7}{R_n} \quad \frac{\text{m}^2}{\text{kg(thrust)}} \tag{3}$$

Since the energy intercepted is about 1 kw/m$^2$, one arrives at 10.8 kw/lb(thrust) at $R_n$ = 1.0 or 12 kw/lb(thrust) at $R_n$ = 0.9. The additional weights of propulsion, structure, such as gondola etc., and hydrogen itself leave very little weight for the collector system. In order to arrive at an overall thrust-to-weight ratio of at least $10^{-2}$, the collector weight (lb) to thrust (lb) must be about 5:1 to 7:1. This then allows about 0.043 to 0.06 lb weight per ft$^2$ intercepted area for the collector system. Such values can obviously not be realized with a parabolic reflector for structural reasons.

During discussions of this problem with the author's associates Messrs. F. D'Vincent, C. Edenfield and O. Dahlke[3)] the proposal was advanced to use a thin-walled, pressure-stabilized sphere. Such a system not only has the least possible weight, but also yields a naturally correct reflector which

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in this case would be a hemispherical reflector. The reflector would be produced by spraying one half of the sphere with a thin metallic film of aluminum or silver, while the other half would remain highly transparent. The resulting weight reduction is so large that a certain increase in intercepted area to compensate for slight absorption losses on the transparent side can easily be accomodated. An analysis of the hemispherical reflector is presented in the Appendix and it is shown that heat transfer rates of maximum practical intensity are obtained along that part of the optical axis which represents the focal line in a reflector of this shape. It was therefore decided to investigate a vehicle design on this basis.

6. The Solar-Powered Space Ship Prototype

For this purpose, a small hydrogen-operated prototype of about 16,000 lb gross weight was assumed, containing about 11,000 liquid hydrogen and a gondola for two crew members. Several configurations were considered. The eventual design which is shown in principle in Figure 8 and in more detail in Figure 9 was chosen, because it appears to be the most attractive compromise from a number of viewpoints.

Gondola, spherical hydrogen tank and exhaust nozzle represent one rigid system, located in the center between two collectors. Center portion as well as reflectors can be rotated freely with respect to each other about the axis of rotation connecting the centers of the two collector spheres. Large vanes in the motor exhaust provide roll control (about the thrust axis). By means of combined motion about axis of rotation and thrust axis, as well as yaw control about the third axis normal to the plane of the paper, complete freedom of orientation of thrust axis and optical axis with respect to each other is assured. Yaw control is effected by means of tilting the exhaust nozzle, using actuating cylinders which also provide a possibility for fine control of the thrust about the axis of rotation by having these actuators operate in two different planes, as explained in Fig. 9, Detail A. Detail B explains the connection between collector sphere and tank sphere and shows that an electric drive is provided for each collector to rotate with respect to the thrust axis. The thrust axis in turn can rotate with respect to the collectors by tilting the motor normal to the plane of the paper. The vehicle can thus accelerate in any direction with respect to the sun. The arrangement of equipment in and around the collector spheres is such that the center of gravity of each collector is in the center of the sphere, or at least on the axis of rotation, so that the reflectors can be rotated without producing a moment arm with respect to the line of thrust.

The flow diagram Fig. 8 shows that hydrogen is pumped from the tank into the collector sphere. This is done by means of a low-pressure pump to keep the flow velocity and therewith the pressure losses down during the about 74 ft long flow to the center of the sphere. If necessary, additional pumps must be installed along the way to maintain the flow at the desired booster pressure prior to entering the high-pressure pump in the center of the collector. Since the flow quantities are small - of the order of 0.1 to 0.2 lb for each collector - and since the pressure is relatively low, the weight of these booster pumps together with their electric drive is very small, of the order of a few pounds. In the high-pressure pump system, the pressure is increased to the order of 30 atm (440 psi), to intensify the heat transfer. Subsequent pressure losses on the way to the motor are not allowed to reduce the pressure below 15 atm, using again booster pumps if necessary. Expansion through the nozzle takes place at an initial pressure of about 15 atm (220 psi). The arrangement of the high-pressure pump close to the heating element reduces the length of high pressure lines. Assuming that the inlet pressure at the high-pressure pump is 30 psi and the outlet pressure 450 psi and assuming further an efficiency of 0.5 for the pump and 0.8 for the electric motor, the horsepower requirement for each of the two systems is about 9 HP or 6.7 kw. This and additional power is probably most economically provided by means of a turbo-electric system. In Fig. 9 a single-stage impulse turbine is indicated prior to the expansion nozzle. By means of an alternator AC is produced ( 3 phase, 4 wires, 120/208 V) which has the advantage of yielding lower transmission losses and permitting a higher rotational speed of the motors and pumps than with DC where the motor speed is limited to about 8,000 to 10,000 rpm. Of the above mentioned 4 wires, 2 each lead to the collector spheres. By keeping them in close contact with the cold hydrogen lines, transmission losses can be minimized. These losses are estimated to be about 0.25 HP per leg.

For starting the propulsion system a high-pressure gas tank for the initial pump operation is indicated in Fig. 9. However, for repeated starts solid propellant starter rockets to energize the turbine and produce a small amount of thrust appear preferable.

Energy for auxiliary power needed in the gondola and for radio purposes can be made available either from radiation of lesser intensity in the heater area or from special solar batteries shown at the extension of the respective optical axes of the collectors in Fig. 9. Storage batteries take over while the vehicle passes through the shadow of the earth. The solar batteries outside the collector spheres would block out radiation arriving in a cylindrical space around the optical axis. It is shown in the Appendix that this radiation furnishes only a negligible contribution to the energy concentration along the optical axis (focal line).

The heater extends from a point half way from the center to a point on the periphery of the reflector. Figure 10 shows the intense heat flux density along the focal line for several values of the reflectivity (cf. Appendix). Peak values between 8 and 10 Btu/in$^2$ sec are reached at $\varphi$ about 45 degrees, corresponding to approximately 70 percent of the distance from the center to the periphery (Appendix).

The spherical collector proper is assumed to consist of polyester (polyethylene terephthalate), a clear and transparent plastic of considerable strength and very light weight. Very thin films of polyester can be manufactured and the design shown in Fig. 9 is based on a thickness of 0.001 inch. Under these conditions the material is not expected to absorb any appreciable amount of light or to produce significant refraction. In order to utilize the industrial state of the art, a polyester called Mylar D which is being produced, has been evaluated (17). The subsequent information, representing average values, is drawn from this reference and is summarized in Table 3. The polyester film transmits about 90

percent of the incident light in the visible region. The tensile strength is satisfactory, even at 300°F (150°C). Under the given conditions of thrust-to-weight it appears sufficient to pressurize the sphere with hydrogen or helium at 0.01 psi (about $7 \cdot 10^{-4}$ atm) to lend adequate rigidity to the hemispherical reflector. This yields a skin stress of 3,840 psi which is well under the tensile strength left at 300°F. This is approximately the daylight temperature of the lunar surface. It is expected that the film temperature will stay below this level, since it absorbs less radiation than the moon's surface. Moreover, since hydrogen or helium will diffuse through the skin, especially on the transparent side, a certain amount of cooling is provided automatically where it is most desirable. The very low pressure on the other hand keeps the amount of gas pressure to be renewed on a very low level. For the same reason also micron-size holes punctured into the film by cosmic dust are not expected to be critical, depending, of course on the density of the dust. To a certain extent the transparent film and the internal gas which in spite of its rarefaction contains more than $10^{20}$ molecules per $ft^3$, provide protection for the reflector proper against frontal impingement by cosmic dust. The rear side is more resistant, since this side is expected to be metal-sprayed on both, the inner and the outer side with a layer about one micron thick. The weight of the metallic film is negligible and the film thickness could be increased if desirable for reasons of protection from cosmic dust. In the direction of low temperature the polyester material shows also good qualities. Mylar is quoted in ref. (17) to be free from embrittlement at temperatures as low as - 60°C (-76°F). The thermal radiation given off by the heater when cooling down and the heat content of the thin internal atmosphere, will greatly dampen the temperature drop normally encountered when the vehicle enters the shadow of the earth.

Thus the material seems to be applicable to this design. However, it is realized that there are many unknowns left, particularly with respect to cosmic dust and the effect of exposure of this material to those parts of the solar spectrum which are absorbed by the atmosphere. However, it appears still too early

to rule out this type of application. Moreover, further directed development may lead to additional improvements. The problems encountered in this design are characteristic problems of an advanced space technology and must be solved jointly by industrial and satellite research.

A weight summary of the prototype is presented in Table 4. Both collectors together weigh only 740 lb or, with accessories, 1,000 lb. This is roughly the equipment which replaces the 44,000 lb oxygen otherwise needed to heat the hydrogen (11,000 lb) chemically. It is this fundamental advantage which makes the solar-powered space ship a significant possibility and well worthwhile the effort to solve its numerous existing problems. Of course, this advantage is not all gain. There are certain penalties in the form of design and operational difficulties. Another, most significant penalty, namely the flight mechanical performance loss due to the low thrust-to-weight ratio, will be discussed in the subsequent section.

## 7. Summary and Conclusions

The basic characteristics of non-chemical propulsion methods are surveyed. The principal reason for a change to these systems is the attempt to get away from the high mass consumption which represents the major disadvantage of the chemical propulsion method. However, it is pointed out that in doing so, a host of other, partly even more severe difficulties is encountered. All non-chemical methods require considerable research and development effort before they can become practical.

Because of its advantages in simplicity and high energy conversion efficiency the solar-powered space ship is discussed in more detail. This vehicle is potentially capable of carrying out operations in cislunar and lunar space (without landing) on a much more economical basis as far as supply requirements are concerned. However, the system is a good example for the manifold difficulties encountered when changing to another propulsion system. These difficulties are not necessarily insurmountable, but their severity is considerable and the resulting problems present many challenges to engineering and science. In the case of the solar-powered space ship the requirement for extremely light construction is the determining factor, to a greater extent than ever before in the history of rocket development which is the story of man's flight against weight in more than one sense. It appears that this problem can be overcome by radically new designs which must be based on the best materials industry can provide. The prototype presented here may indicate a possible solution. It appears likely that if at a later time industrial and scientific ingenuity can be put to work on a broader basis, a practical solar-powered space ship can be developed and man's freedom of operation in the earth-moon field be increased decisively.

Appendix : Analysis of the Spherical Reflector

Figure // defines the notation used subsequently. The optical or principal axis passes through the vertex V and the center O of the sphere. For the sake of simplicity of the geometrical relations involved, solar radiation is in this connection assumed to consist of parallel rays of light. The reflection point R of an incoming light ray can be described by the center angle $\varphi$ , measured from a line normal to the incoming light and into the direction of the optical axis toward the vertex V. It is then

$$\frac{h}{r} = \cos\varphi \qquad (1\text{-}1)$$

$$\frac{h}{f} = \sin 2i$$

and

$$\frac{f}{r} = \frac{\cos\varphi}{\sin 2\varphi} \qquad (1\text{-}2)$$

The last relation defines the length f of the reflected beam in terms of the radius of the spherical reflector. This length, times the angles subtended by the light source, determine the diameter of the image. In the case of the sun this angle is known to be 32' = 0.00931 radians = $\beta$ . The smallest possible image diameter is therefore $f\beta$ at any station of the optical axis irradiated, and it lies in the plane normal to the direction of f. If the element lies in the axis VO, the incident radiation will spread to form an ellipse of light rather than a circle. The major axis of this ellipse is given by

$$2a = \frac{f\beta}{\cos(90-i)} = \frac{f\beta}{\sin 2\varphi} \qquad (1\text{-}3)$$

The radiation intercepted by the minimum area $A_{min} = (\pi/4)f^2\beta^2$ , as compared to the total elliptic area irradiated is,

Elliptic area $A_{ell} = a\,b = \pi\,\frac{1}{2}\,\frac{f\beta}{\sin 2\varphi}\cdot\frac{1}{2}\,f\beta = \frac{\pi}{4}\,\frac{f^2\beta^2}{\sin 2\varphi}$

$$\frac{A_{min}}{A_{ell}} = \sin 2\varphi \qquad (1\text{-}4)$$

Assuming a flat plate receiver, the radiation absorbed is (for $\alpha = \epsilon$)

$$\alpha = \epsilon = \epsilon_n \cos(90-2i) = \epsilon_n \sin 2i \qquad (1\text{-}5)$$

For a small reflector surface element the incident flux is then given by

$$q = S \cos i = S \sin\varphi \qquad (1\text{-}6)$$

where S is the solar constant. This reduction in intensity is of course due to the spreading of the incident flux over a larger area than in the case of vertical incidence (at point V). The reflectivity of the surface element follows from Kirchhoff's law (assuming $\alpha = \epsilon$), $2\epsilon + R = 1$; $\epsilon = (1-R)/2$ and Lambert's law $\epsilon = \epsilon_n \cos i = \epsilon_n \sin\varphi$, whence

$$R = 1 - (1-R_n)\sin\varphi \qquad (1\text{-}7)$$

The amount of radiation absorbed by a flat plate receiver (diameter $f\beta$) from a surface element af arc length ds, revolved about the optical axis, is therefore

$$dq = S \sin\varphi \; R \; \frac{A_{min}}{A_{ell}} \; \alpha \; 2\pi \,(2h)\; f \, d\varphi \qquad (1\text{-}8)$$

where $S \sin\varphi$ is the incident flux, R the reflectivity, $A_{min}/A_{ell}$ the fraction of radiation intercepted by the minimum area, $\alpha$ the absorptivity, $2\pi(2h)$ the circumference at the given h about the optical axis and $f\,d\varphi$ the arc element. Using Eqs. (1-7), (1-4), (1-5) and (1-1) and setting $f\,d\varphi = ds$, one obtains the flux density per area element

$$\frac{dq}{ds} = 4\pi r S \epsilon_n f_1(\varphi)\left[1-(1-R_n)\sin\varphi\right] \qquad (1\text{-}9)$$

where

$$f_1(\varphi) = \sin^2 2\varphi \sin\varphi \cos\varphi \qquad (1\text{-}10)$$

Integration of this equation yields the overall flux from the reflector surfac on the irradiated portion of the optical axis,

$$Q = 8 \ r^2 S \epsilon_n \left[ \frac{1}{5} \left( \sin^3\varphi \cos^2\varphi + \frac{2}{3} \sin^3\varphi \right) - \frac{1-R_n}{6} \left( \sin^4\varphi \cos^2\varphi + \frac{1}{2} \sin^4\varphi \right) \right] \quad (1\text{-}11)$$

The flux density in terms of the minimum area $A_{min}$ is at any point

$$q = \frac{Q}{\frac{\pi}{4} f^2 \beta^2} = \frac{32 \ S \epsilon_n}{\frac{f^2}{r^2} \beta^2} \left[ f_2(\varphi) - \frac{(1-R_n)}{6} f_3(\varphi) \right] \quad (1\text{-}12)$$

where

$$f_2(\varphi) = \frac{1}{5} \left( \sin^3\varphi \cos^2\varphi + \frac{2}{3} \sin^3\varphi \right) \quad (1\text{-}13)$$

$$f_3(\varphi) = \sin^4\varphi \cos^2\varphi + \frac{1}{2} \sin^4\varphi \quad (1\text{-}14)$$

Eqs. (1-2), (1-4), (1-10),(1-13) and (1-14) are tabulated in Table 5. The tabulation begins with $\varphi = 30°$, because at this value the incoming ray hits the vertex after reflection. At smaller values of $\varphi$ the ray follows a polygon path before it hits the optical axis. No attempt has been made in this first survey to analyze this portion of the incoming radiation. Qualitatively, it increases the radiation influx in the outer portion of the focal line ($0.7 \leq r \leq 1.0$).

The ratio f/r as well as dq/ds are plotted in Fig. 12 . The curve f/r shows the well-known fact that for spherical reflectors of very small aperture a focal point exists which is located at 0.5 r. The curve dq/ds indicates that for large aperture spherical reflectors the maximum differential radiation flux is at $\varphi = 45°$, corresponding to 0.7 r.

Using Eq. (1-11) the integral radiation flux can be computed. The result is shown in Figure 13 , where Q is given in terms of $\pi r^2 S \epsilon_n$ as function of $\varphi$ . In accordance with the trend shown by the dq/ds curves, the Q curves indicate that there is little or no contribution from radiation influx at the extreme values of $\varphi$ . This trend is the same as found in (16) for the parabolic mirror.

By computing average values of q, Eq. (1-12), one can find the temperature of a black body receiver (flat) along $0.5 \leq r \leq 1.0$, by using the Stefan-Boltzmann relation $T = (q/\sigma)^{\frac{1}{4}}$ where $\sigma$ is the Stefan-Boltzmann constant. This has been done by using average values of q in intervals of five degrees. The result is shown in Fig. 14 for the radiation flux density per $cm^2$ of minimum area and for the temperature. This temperature, representing the highest theoretically attainable in a reflector of this type ($R_n = 1.0$), is 1,000°K or 1,800°R lower than the theoretical maximum found in (16) for the parabolic mirror. The values of q are also presented in Fig. 10 where they are presented in the engineering system.

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TABLE 1 SOME CHARACTERISTICS OF PROPULSION SYSTEMS

| Characteristics | Chemical Combustion | Pile-Heating | Solar-Heating | Arc-Heating | Electric (Ions) |
|---|---|---|---|---|---|
| Energy Conversion Sequence | Chemical<br>Thermal<br>Kinetic | Nuclear<br>Thermal<br>Kinetic | Solar<br>Thermal<br>Kinetic | [Solar or Pile or $\beta$ - Decay]<br>Electric<br>Thermal<br>Kinetic | [Solar or Pile or $\beta$ - Decay]<br>Electric<br>Kinetic |
| Range of Specific Impulses (lb sec/lb) | 200-300 (surface)<br>300-400 (space) | 400-700 ($T_c$= 2,000°K)<br>600-900 ($T_c$= 3,000°K) | 400-500<br>($T_c \leq$ 1,100°K) | 700-1,200<br>(Depending on Design) | $\geq$ 13,000<br>(Depending on Design) |
| Systems Specific Thrust (lb thrust/lb prop. syst. hardware) | 50 - 80<br>(Engine, Pump, Pressurization Thrust Structure Auxiliary System) | Probably less but same order of magnitude (without shielding), Depends on engine size | $\sim$0.05 | $\sim$0.05 ($\beta$- Decay)<br>0.005-0.0015 (Solar)<br>(Depending on Design) | $\sim$0.007 |
| Specific Energy Consumption (kw/lb thrust) | 0.05-0.025 (Prop. Syst.)<br>6.5-13 (Converted to heat of medium)<br>4-8 (Converted in Jet) | 15-20 (Pile)<br>$\sim$ 14-18 (Converted to to heat of medium)<br>10-15 (Converted in Jet) | 12-14 (Propulsion Syst)<br>11-13 (Based on Mirror Size R = 0.<br>9-11 (Converted to heat of medium)<br>8-10 (Converted in Jet) | 60-90 (Propulsion Syst)<br>20-30 (Converted to heat of medium)<br>14-21 (Converted in Jet) | $\sim$ 300 (Converted in Jet) |
| Acceleration (Thrust/Wt) | 1.2-1.5 (Surface)<br>$\geq$ 0.25 (Space) | → | $\sim 10^{-2}$ | $5.10^{-4}$ - $5.10^{-3}$ | $10^{-5}$ - $5.10^{-5}$ |
| Working Fluid (Medium) | Fuel & Oxidizer | $NH_3$, $H_e$, $H_2$ | $H_2$ | Reducing medium<br>$H_2$<br>$CH_4$, $N_2$ | Cesium, Rubidium or other Ions |
| Maneuverability | Excellent | Excellent | Poor | Poor | Very Poor |
| Areas of Operation | Surfaces<br>Space (Moon, Mars, Venus) | Surfaces<br>Space (Venus-Mars)<br>Conditionally (Saturn) | Cislunar and Lunar Space (Outside Shadow)<br>Inner Solar System | Cislunar and Lunar Space<br>Inner Solar System | Interplanetary Space (inner and outer solar system) |
| Probability of Extra-Terrestrial Supply of Medium. | None | Possibly outer solar system (Titan) | None | None | The best media (lowest ionization potential) not readily available extra-terrestrially |
| General Conclusions:<br>(a) Initial investment in spaceborne hardware in relation to thrust produced | Low | Fairly Low | Fairly High | High | Very High |
| (b) Running Supply in working Fluid (Medium) | Very High | Fairly High | Fairly High | Low | Very Low |
| (c) Propulsion System Maintenance & Reliability | Very limited life of engines & pumps: comparatively complex | Long life, Removal of fission products necessary | Long life; simple syst. Maint. depends on density of meteoritic matter. | Cannot be judged reliably at present | Long life; complex system |
| (d) Promising Areas of Long Range Significance.<br>($\alpha$) Routine | Ascent, Landing (airless) | Ascent, Landing (airless) Lunar-Mars-Venus Transfer | Inner solar system (?) | Cislunar<br>Inner solar system (?) | Interplanetary |
| ($\beta$) Occasional | Mars-Venus-Transfer | Jupiter-Saturn-Transfer | — | — | — |

Table 2 <u>Comparison of Hydrogen and Helium</u>

| | Hydrogen | Helium |
|---|---|---|
| Molecular Weight | 2.016 | 4.003 |
| Freezing Point | -436°F (12)<br>-260°C | -459.4°F<br>-273°C (13) |
| Boiling Point (760mm) | -423°F(12)<br>-252.8°C | -452.2°F<br>-269°C (13) |
| Sp. Gravity (liq.,760mm) | 0.071<br>(4.43 lb/ft$^3$)(13) | 0.13 (13)<br>(8.1 lb/ft$^3$) |
| Critical Pressure | 13.5 atm (12)<br>12.8 atm (13) | 2.28 atm (13) |
| Critical Temperature | -400°F (12)<br>-240°C | -450.2°F<br>-267.9°C (13) |
| Critical Sp.Gravity | 0.031 (13)<br>(1.94 lb/ft$^3$) | 0.069 (13)<br>(4.3 lb/ft$^3$) |
| Specific Heat<br>(Average between B.P. and 3000°k) | 3.7 Btu/lb°F<br>3.7 cal/g°k(5) | 1.13 Btu/lb°F<br>1.13 cal/g°k (5) |
| Heat of Vaporization (760mm) | 108 cal/g<br>194 Btu/lb (12) | 4.93 cal/g (4.93°k) (14)<br>8.87 Btu/lb |

Table 3 Some Relevant Properties of the Polyester MYLAR (17)

| | |
|---|---|
| Melting Point | 250-255°C (about 490°F) |
| Specific Gravity | 1.38-1.39 |
| Thermal Conductivity | $3.63\cdot10^{-4}$ cal/cm sec °C |
| Thermal Coefficient of Linear Expansion | $20\cdot10^{-6}$ °F |
| Light Transmission | about 90% in visible (> 4,000 Å)<br>zero at < 3,000 Å |
| Tensile Strength | 25,000-30,000 psi (about 0°F)<br>10,000 psi @ 300°F (150°C) |
| Bursting Strength | 45 lb @ 1 mil thickness |
| Oxygen Permeability | 0.9 g/100 $m^2$ hr @ 1 mil thickness |
| Hydrogen Permeability | No Data Available |

Table 4 Characteristic Data of the Solar Powered Spaceship Prototype

## 1. Weights

| | | |
|---|---|---|
| Radiation Collectors (2) | | 1,000 lb. |
| Polyester Spheres (2) | 740 lb. | |
| Cold Tubes, Wires, Springs | 190 | |
| Connections, Reflector Drive, Misc. | 70 | |
| Hot Piping System (for 2 Collectors) | | 700 |
| Heating Elements | 200 lb | |
| Hot Piping | 350 | |
| Radiation Jacket for Hot Pipes | 150 | |
| Engine | | 500 |
| Turbine and Alternator | 60 lb. | |
| Low-Pressure Booster Pump (0.37 lb/sec $H_2$) | 10 | |
| Hi-Pressure Pump & Motor (2 sets) | 50 | |
| Wiring (2 sets) (electrical) | 20 | |
| Motor | 50 | |
| Connections & Shut-off Valves | 10 | |
| Array of Solid Prop. Starter Rockets | 300 | |
| Liquid Hydrogen Tank (17 ft Dia., 900 $ft^2$ Surface) | | 800 |
| Controls, Crew & Equipment | | 2,400 |
| Liquid Hydrogen | | 11,000 |
| Gross Weight | | 16,400 lb |
| Dry Weight | | 5,600 lb |

## 2. Miscellaneous Data

| | |
|---|---|
| Collector Sphere: Diameter | 128 ft. |
| Intercepted Area, A. | 12,870 $ft.^2$ |
| Circumference | 402.1 ft. |
| Surface Area | 51,468.8 $ft^2$ |
| Volume | 1,098,000 $ft^3$ |
| Hydrogen gas wt. in sphere (o.ol psi) 300°F | 2.74 lb |
| -150°F | 6.85 |
| Helium gas wt. is roughly twice that of $H_2$ | |
| Intercepted area theoretically required to produce 80 lb. of thrust at $I_{sp}$ = 450 sec | 9,300 $ft^2$ |
| Reflector efficiency 9,300/12,870 | 0.725 |
| Theoretically produced thrust per reflector | 111 lb |
| Energy theoretically collected by reflector | 1,287 kw |
| Theoretical specific energy consumption | 12.9 kw/lb thrust |
| Actual thrust assumed to be produced per collector | 80 lb |
| Total thrust produced | 160 lb |
| Thrust-to-weight ratio: initial | $0.976 \cdot 10^{-2}$ |
| maximum final | $2.963 \cdot 10^{-2}$ |

## 3. Ideal Performance

| | |
|---|---|
| Loading Factor (L.$H_2$/Gross Wt.) | 0.67 |
| Maximum Possible Mass Ratio Based on Loading Factor | 3.03 |
| Operational Specific Impulse Assumed | 450 lb sec/lb |
| Ideal Velocity Based on Above Data | 15,730 ft/sec |

The ideal performance characteristics indicate that this prototype is not [illegible] circumlunar flight and return into the orbit of departure, taken at 300 n.mi. [illegible]. [illegible] purpose of the prototype is to study problems of design, and [illegible] basic characteristics of the vehicle.

Table 5 Characteristic Functions of the Spherical Reflector

| $\varphi$ (deg) | f/r (-) | $f_1(\varphi)$ (-) | $f_2(\varphi)$ (-) | $f_3(\varphi)$ (-) |
|---|---|---|---|---|
| 30 | 1.0 | 0.3248 | 0.03541 | 0.07812 |
| 35 | 0.8717 | 0.4149 | 0.05048 | 0.12675 |
| 40 | 0.7778 | 0.4775 | 0.06658 | 0.18554 |
| 45 | 0.7071 | 0.5000 | 0.08250 | 0.25000 |
| 50 | 0.6527 | 0.4775 | 0.09708 | 0.31441 |
| 55 | 0.6104 | 0.4149 | 0.10945 | 0.37325 |
| 60 | 0.5773 | 0.3248 | 0.11907 | 0.42187 |
| 65 | 0.5517 | 0.2248 | 0.12585 | 0.45785 |
| 70 | 0.5321 | 0.1327 | 0.13005 | 0.48107 |
| 75 | 0.5176 | 0.0625 | 0.13224 | 0.49358 |
| 80 | 0.5077 | 0.0200 | 0.13310 | 0.49866 |
| 85 | 0.5019 | 0.00261 | 0.13332 | 0.49990 |
| 89 | 0.500 | | | |
| 90 | — | 0 | 0.13339 | 0.50000 |

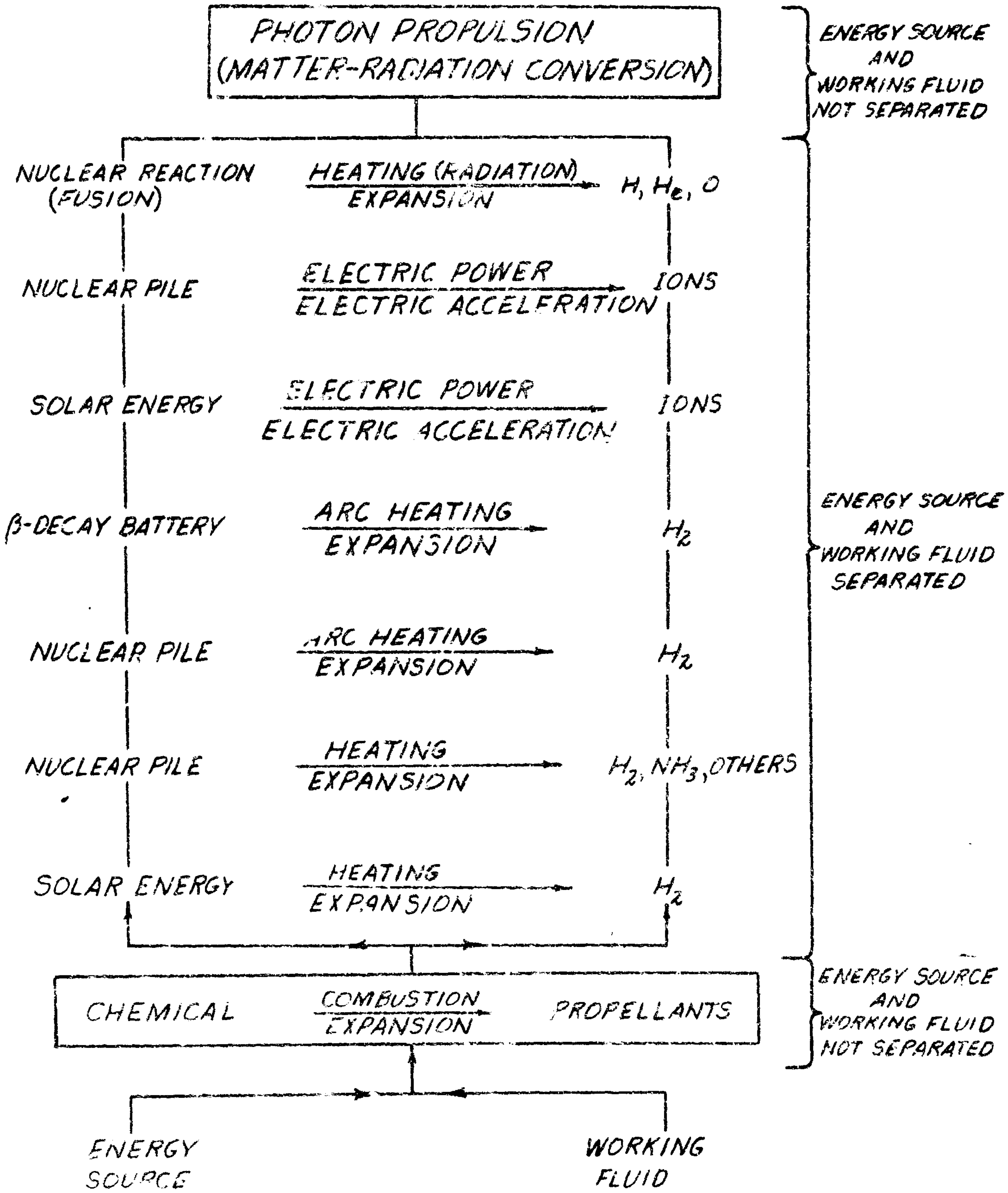



FIG. 1 CORRELATION BETWEEN ENERGY SOURCE AND WORKING FLUID IN VARIOUS PROPULSION SYSTEMS

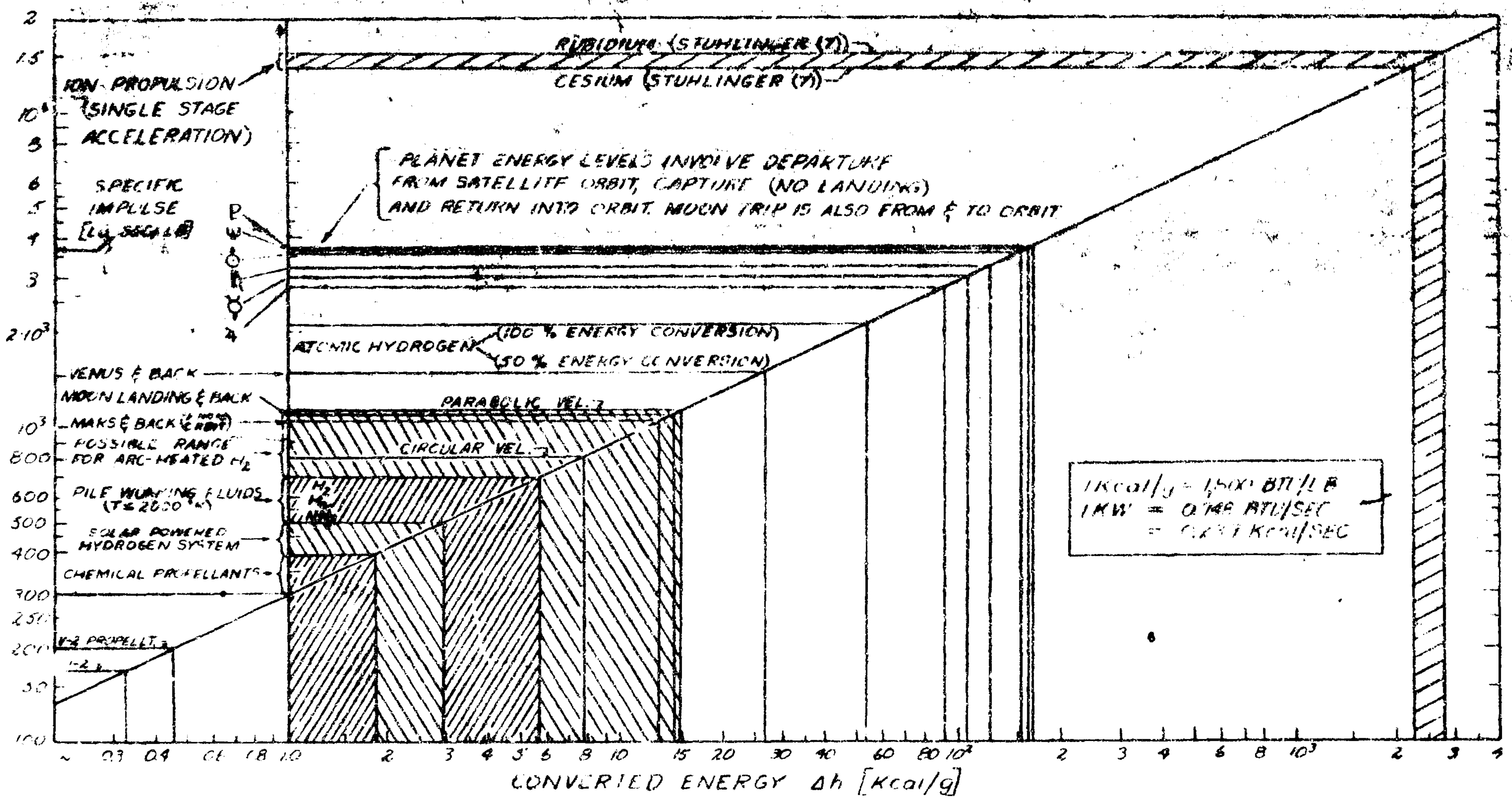
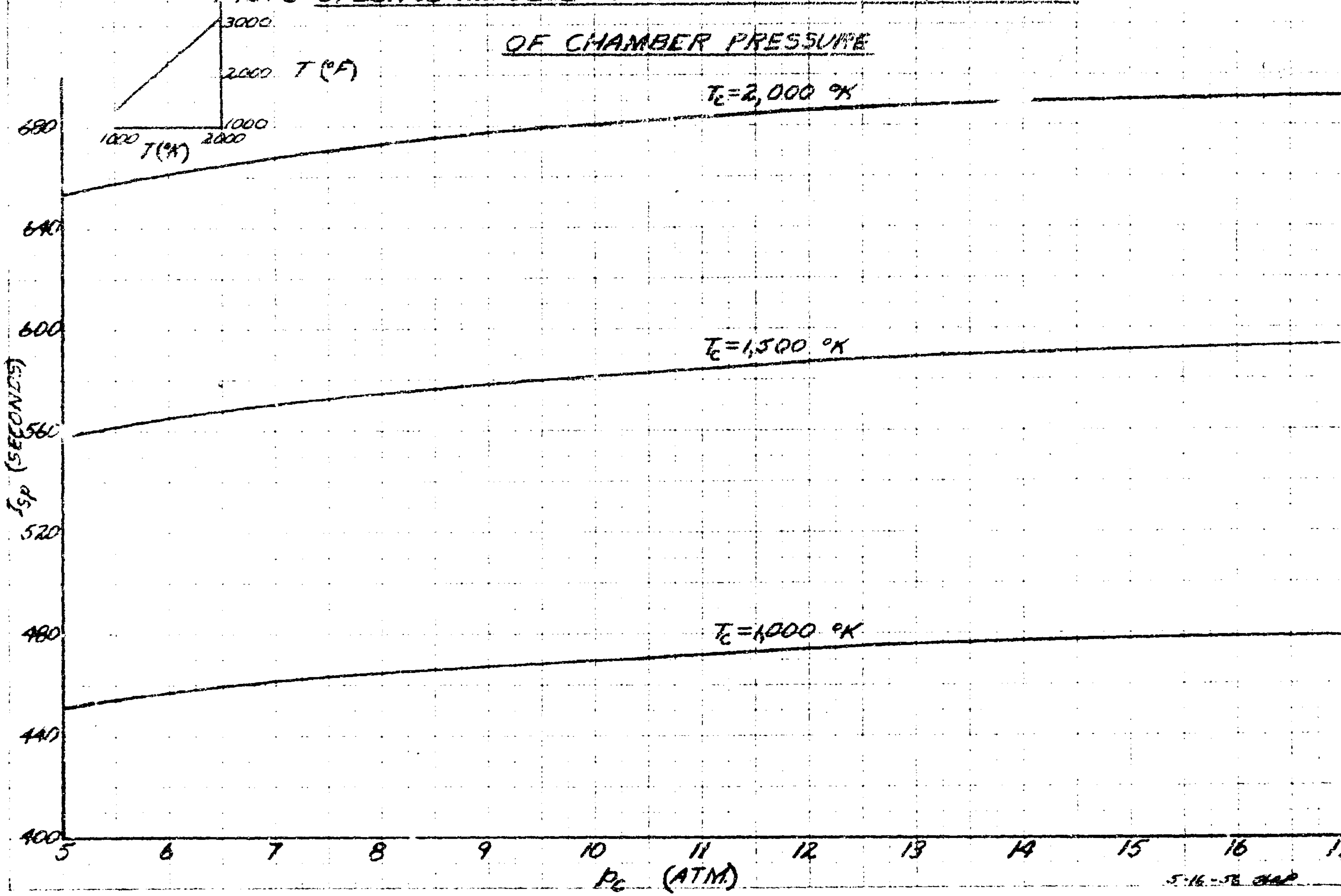



FIG. 2 ENERGY SPECTRUM OF PROPULSION SYSTEMS AND FLIGHT PERFORMANCE

FIG. 5 SPECIFIC IMPULSE OF PURE HYDROGEN AS FUNCTION OF CHAMBER PRESSURE



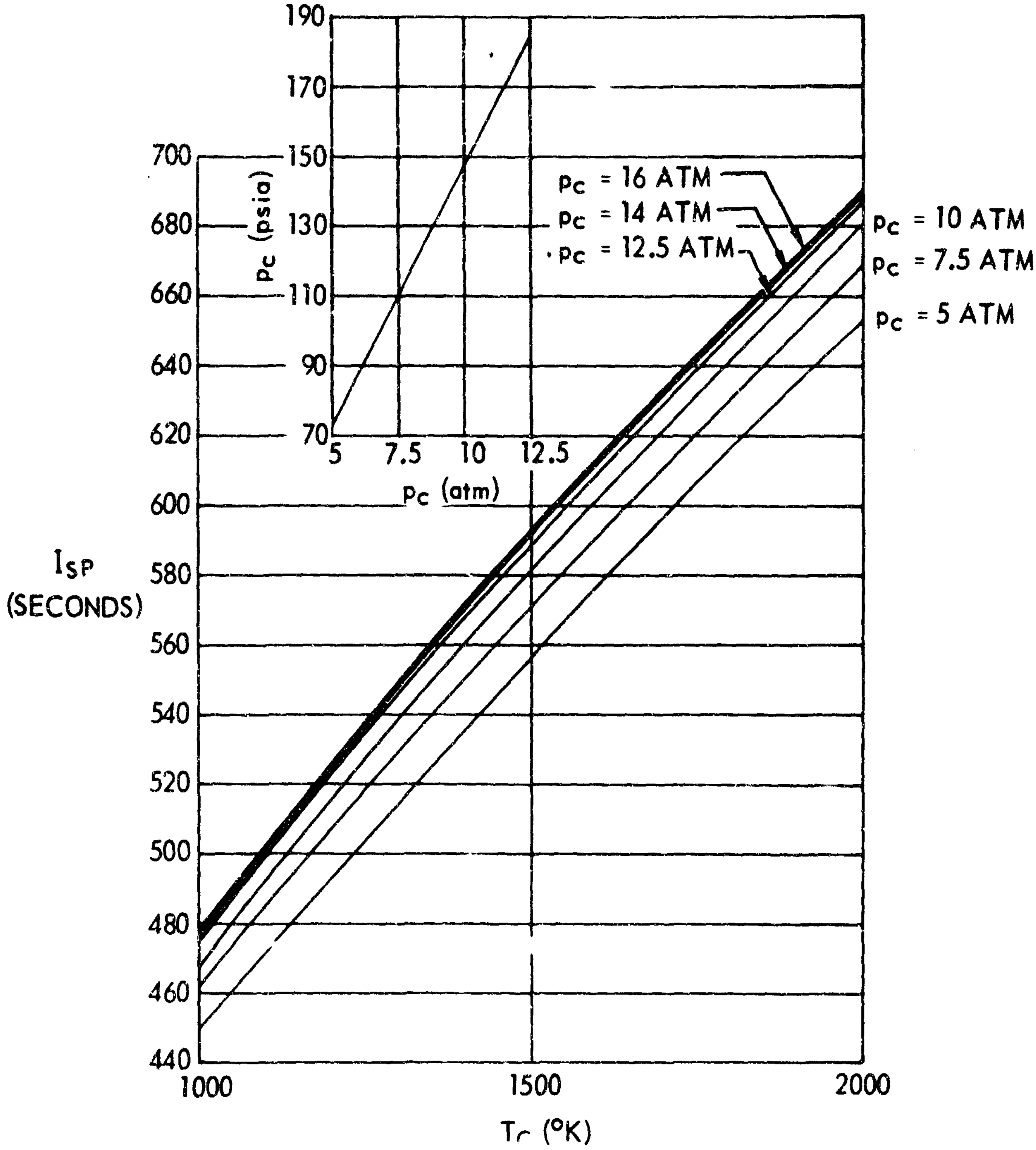
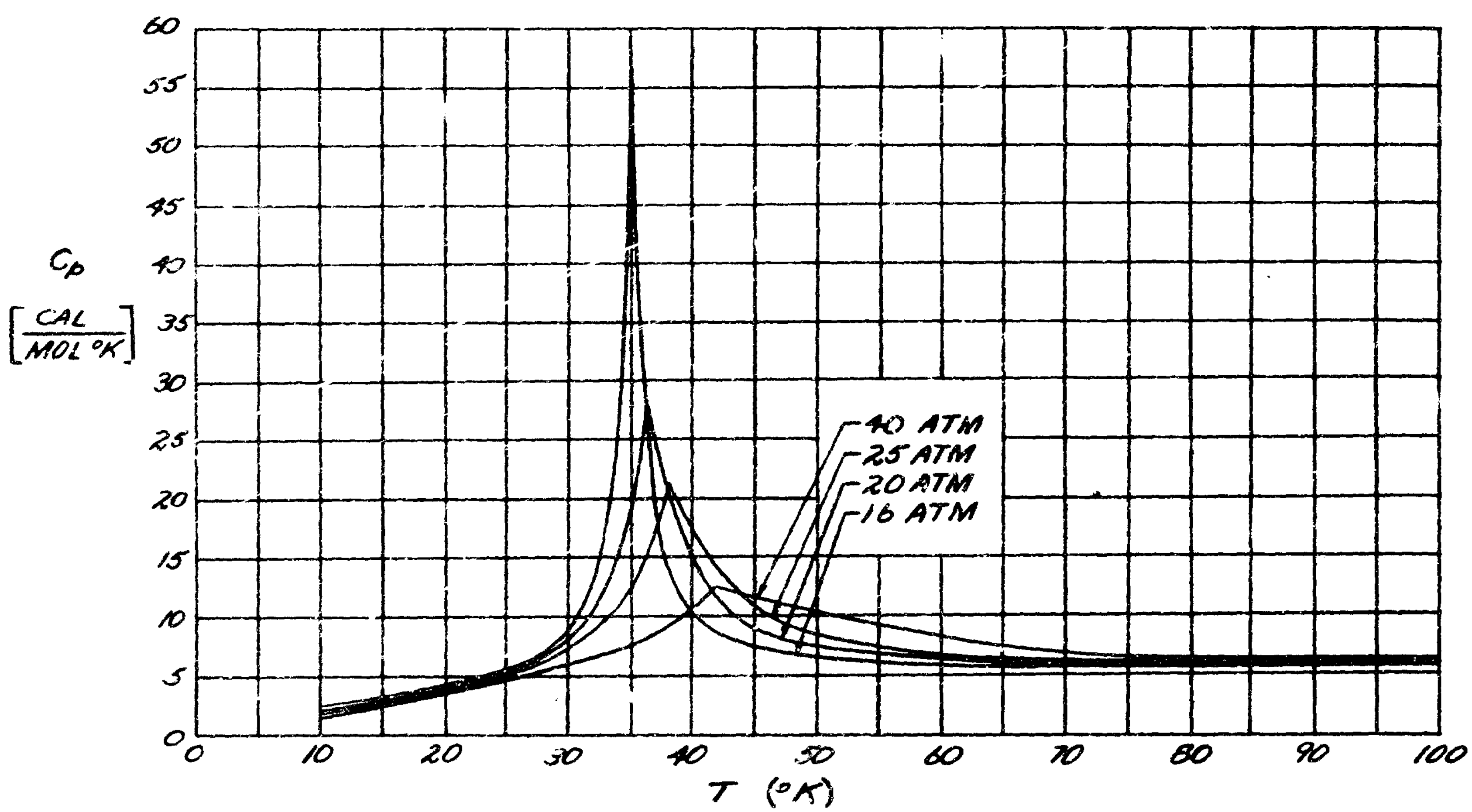



FIGURE 4. SPECIFIC IMPULSE OF PURE HYDROGEN AS FUNCTION OF CHAMBER TEMPERATURE

FIG. 5 MOLECULAR HEAT OF LIQUID HYDROGEN AT SUPERCRITICAL PRESSURE



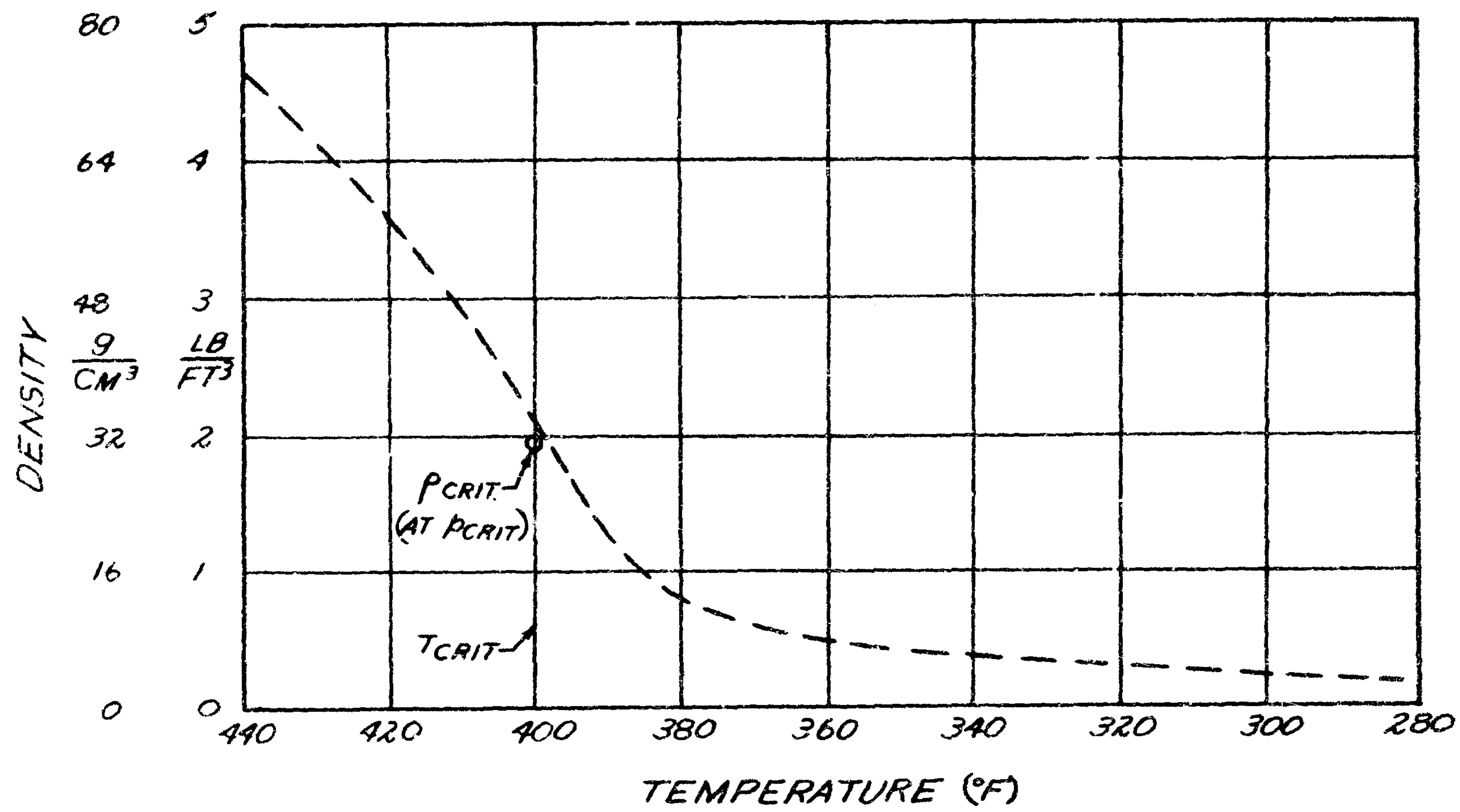
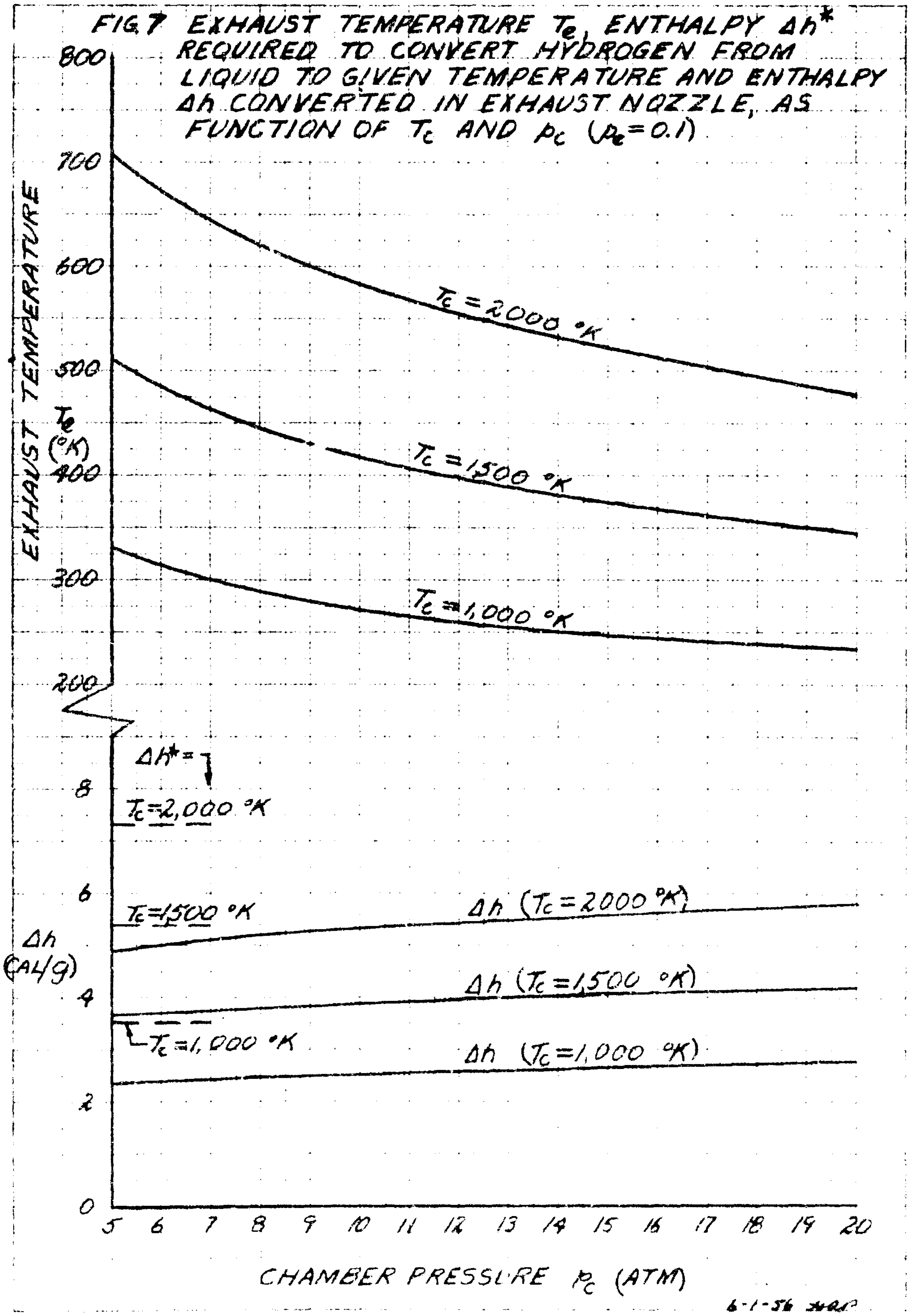



FIG. 6 ESTIMATED DENSITY OF HYDROGEN IN SUPERCRITICAL STATE (15 ATM).

FIG. 7 EXHAUST TEMPERATURE $T_e$, ENTHALPY $\Delta h^*$ REQUIRED TO CONVERT HYDROGEN FROM LIQUID TO GIVEN TEMPERATURE AND ENTHALPY $\Delta h$ CONVERTED IN EXHAUST NOZZLE, AS FUNCTION OF $T_c$ AND $p_c$ ($p_e = 0.1$)



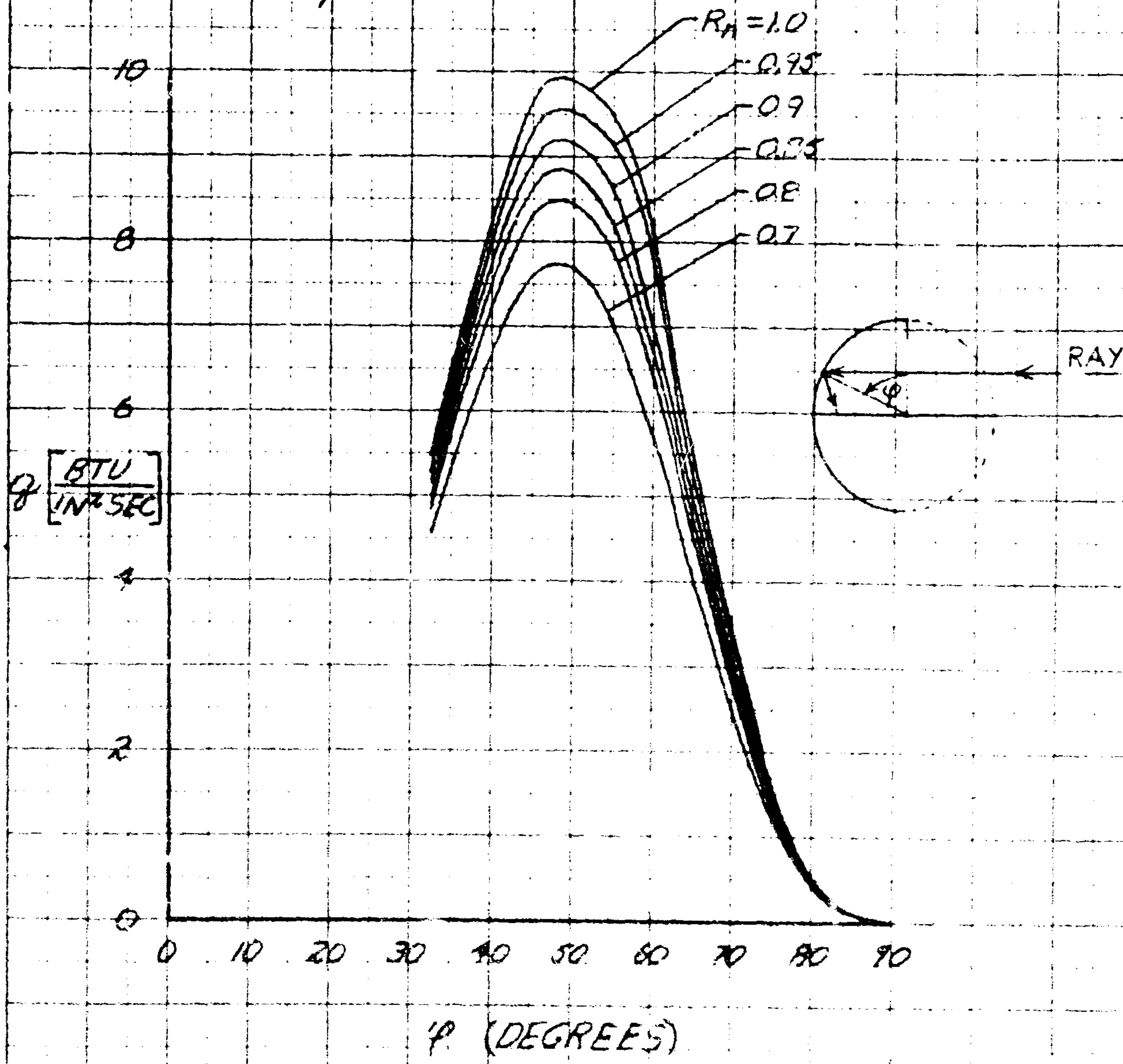

6-1-56

FIG.10 ENERGY CHARACTERISTICS OF SPHERICAL MIRROR (IN SPACE)

RADIATION FLUX DENSITY PER $IN^2$ OF MINIMUM AREA $A_{MIN}$ ALONG OPTICAL AXIS AS FUNCTION OF $\varphi$

(FLAT PLATE RECEIVER)

$$A_{MIN} = \frac{\pi}{4} f^2 \beta^2$$

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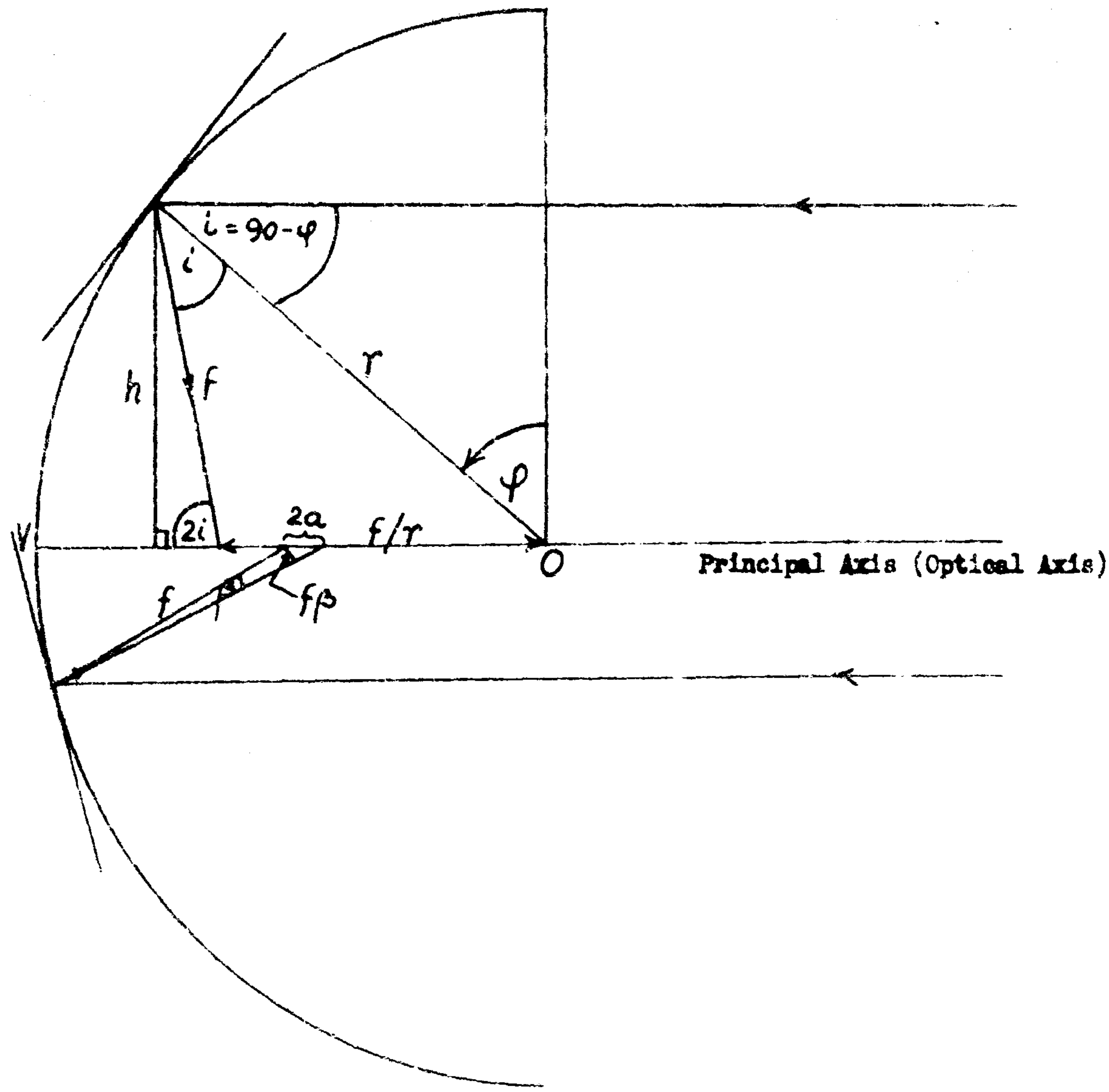



FIGURE 11 SPHERICAL REFLECTOR

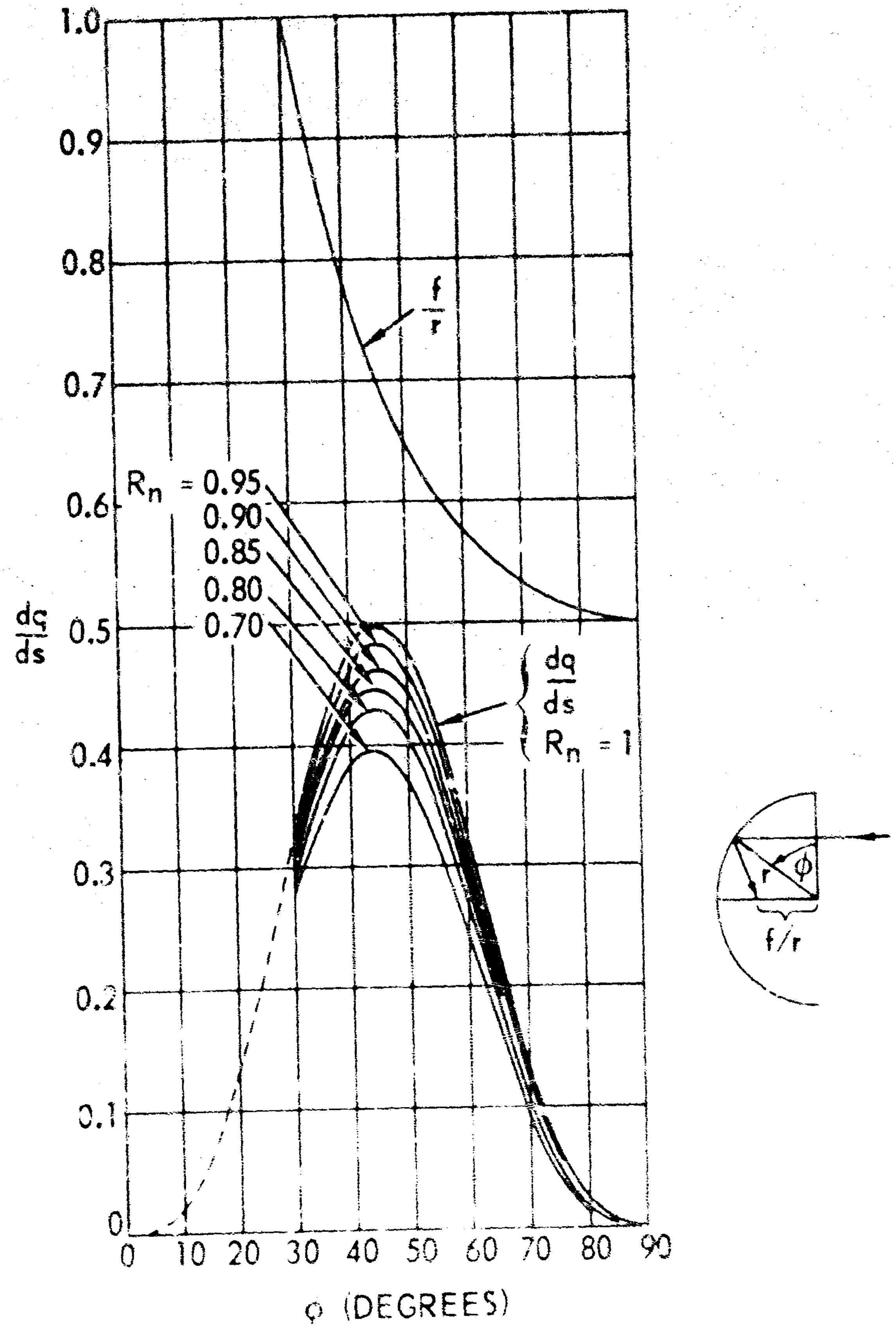
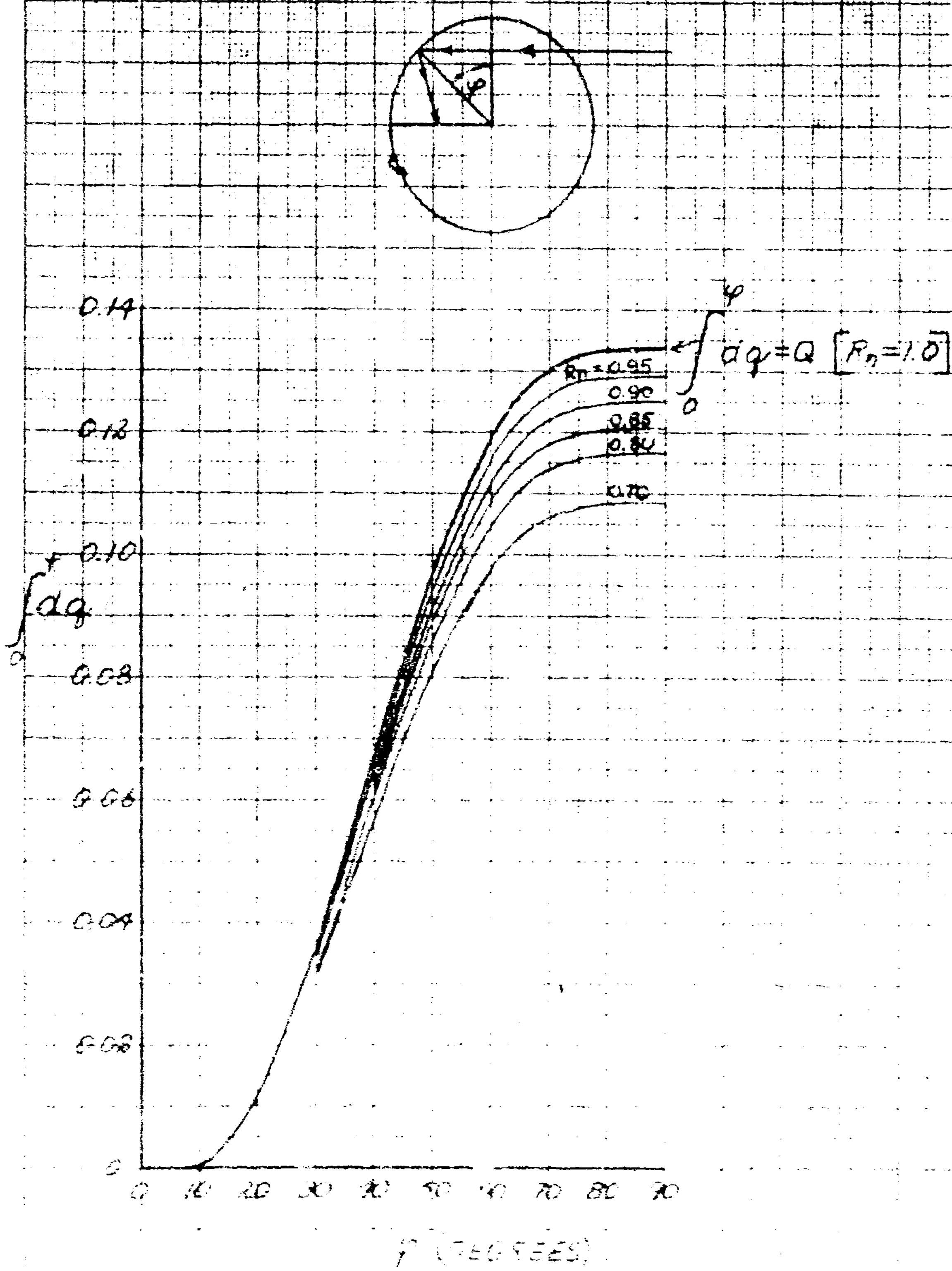
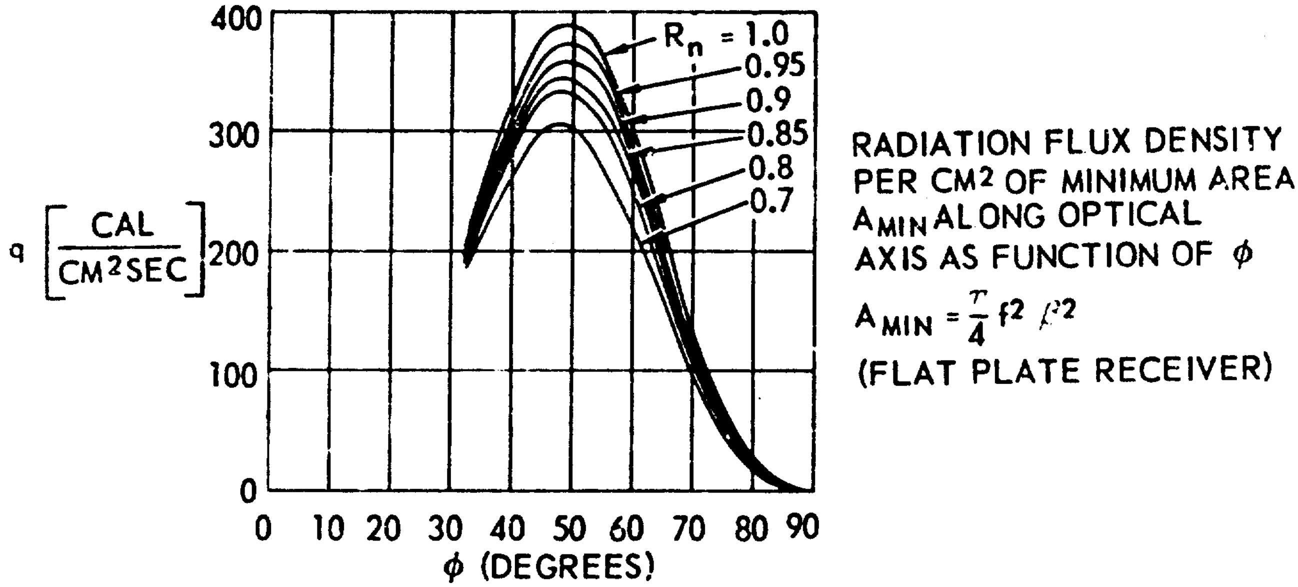



FIGURE 12. HEMISPHERICAL MIRROR:
DISTANCE OF LOCAL FOCAL POINT FROM CENTER
OF SPHERE & DIFFERENTIAL RADIATION FLUX $\frac{dq}{ds}$

FIG 13 HEMISPHERICAL MIRROR:

TOTAL RADIATION FLUX ON FLAT PLATE RECEIVER





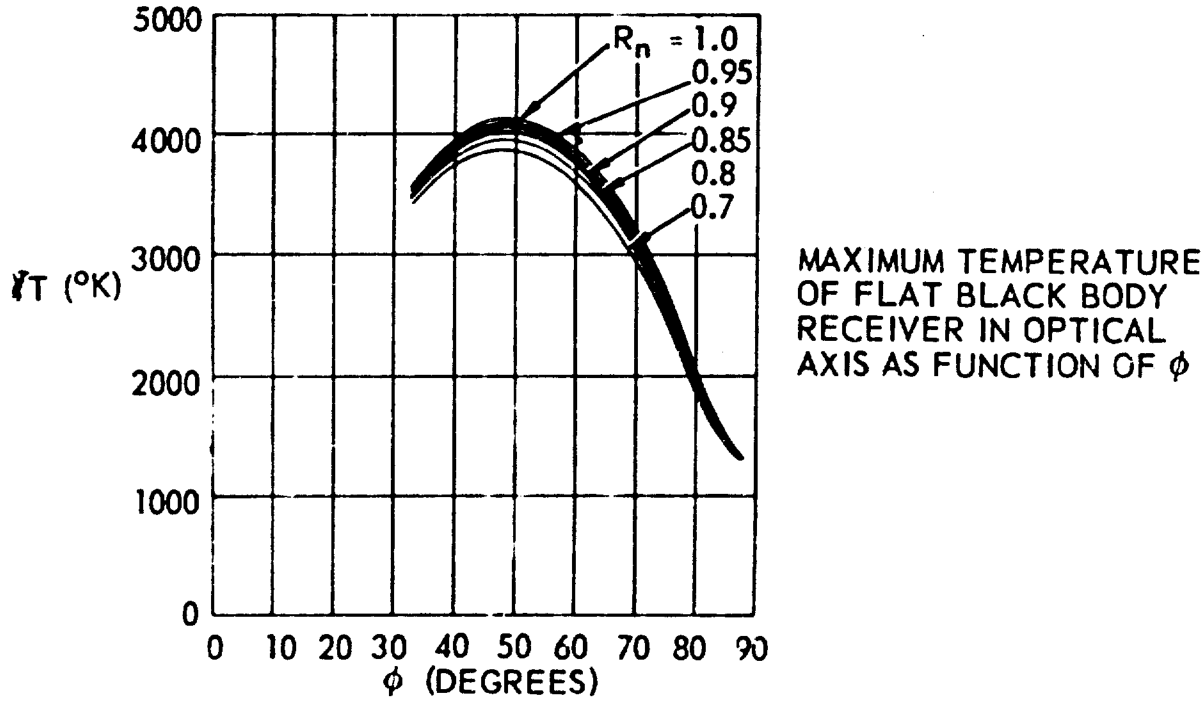
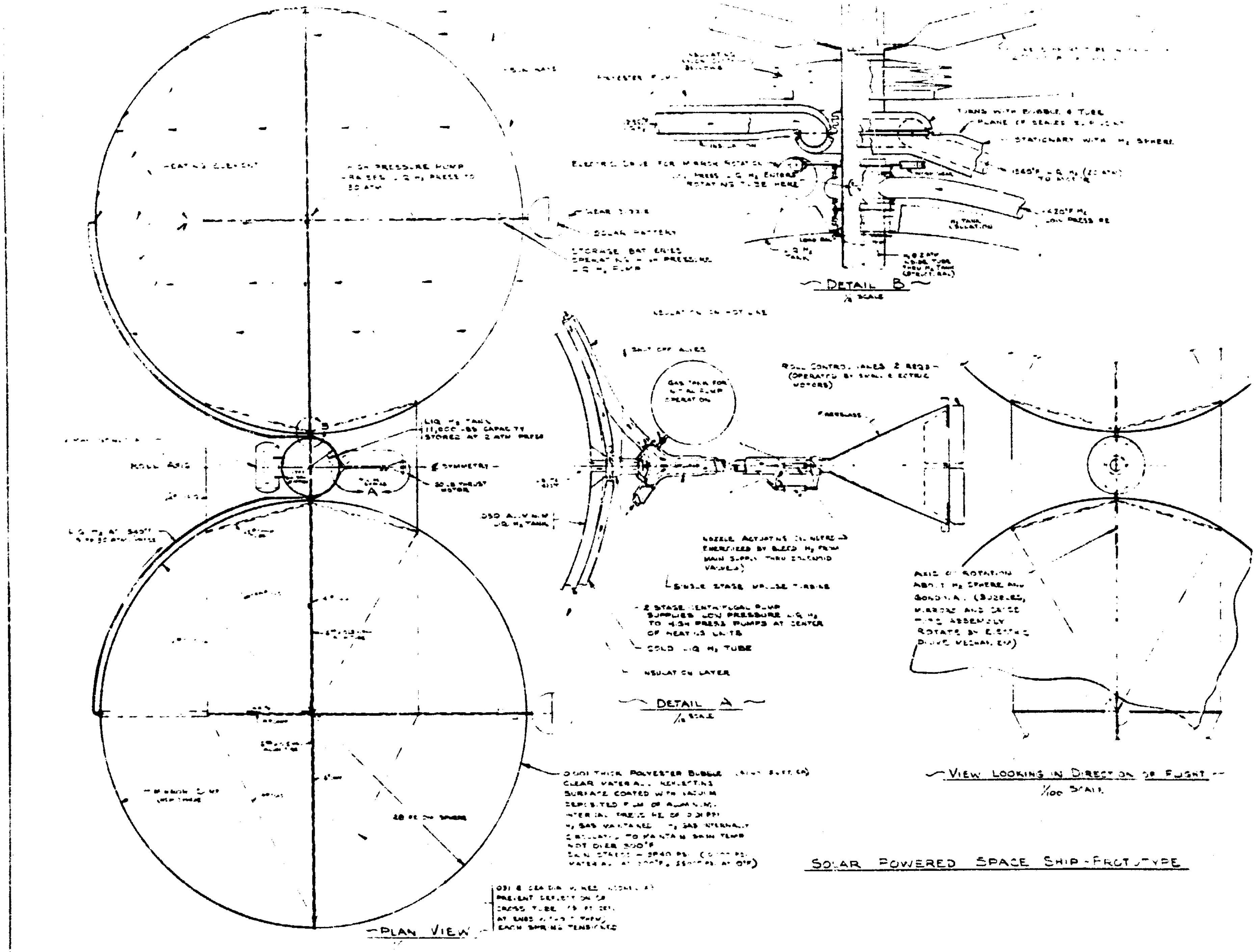
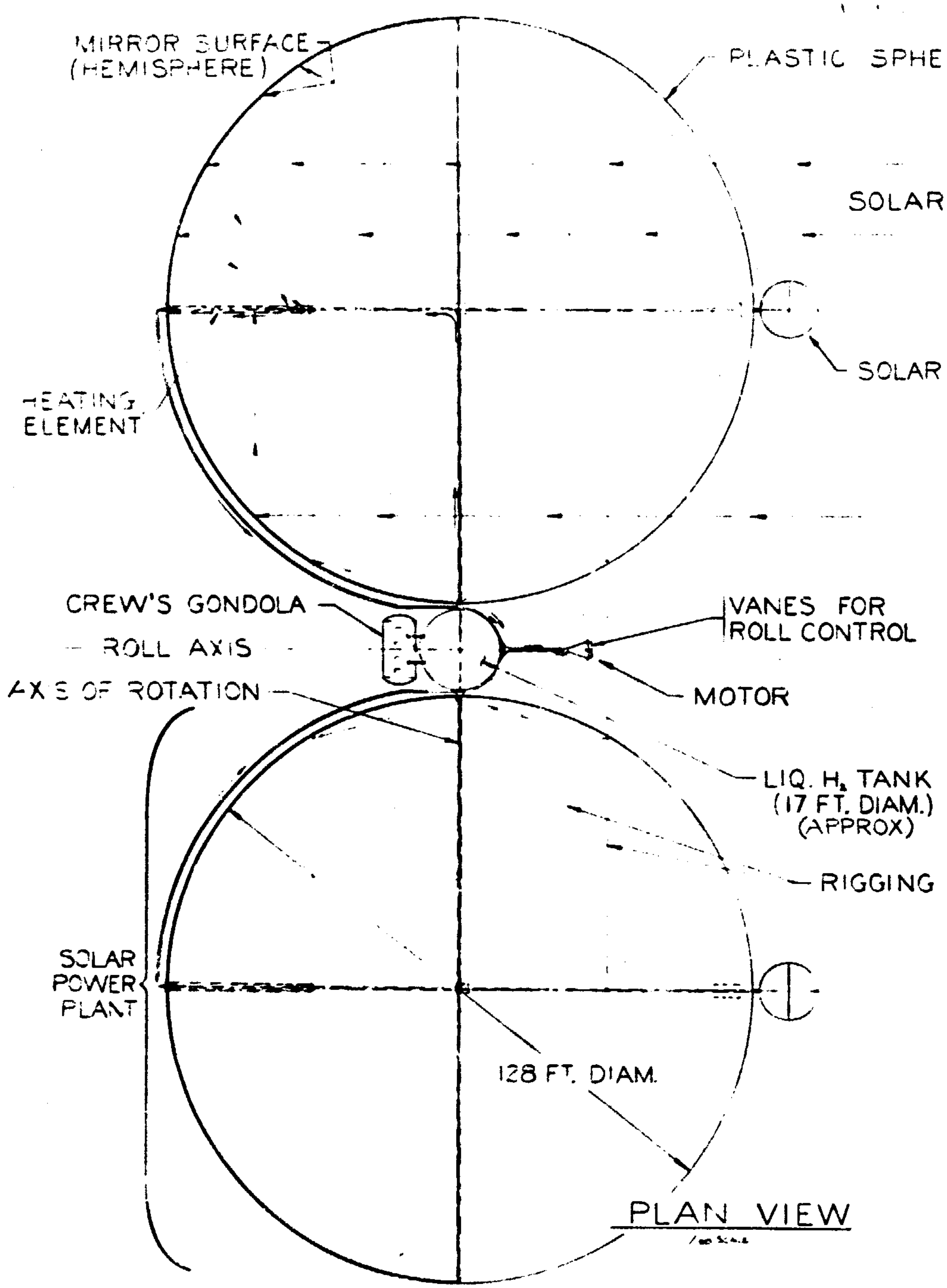
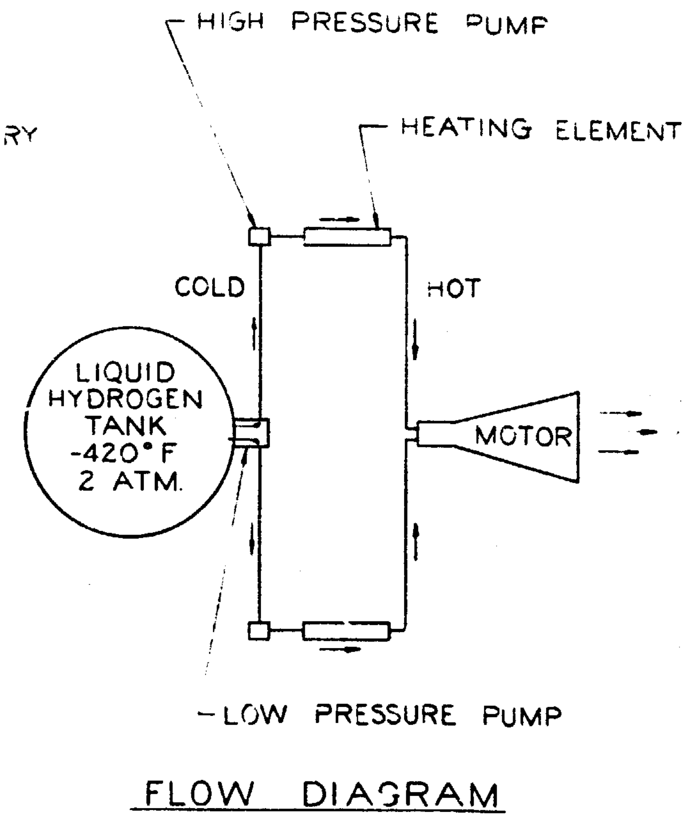



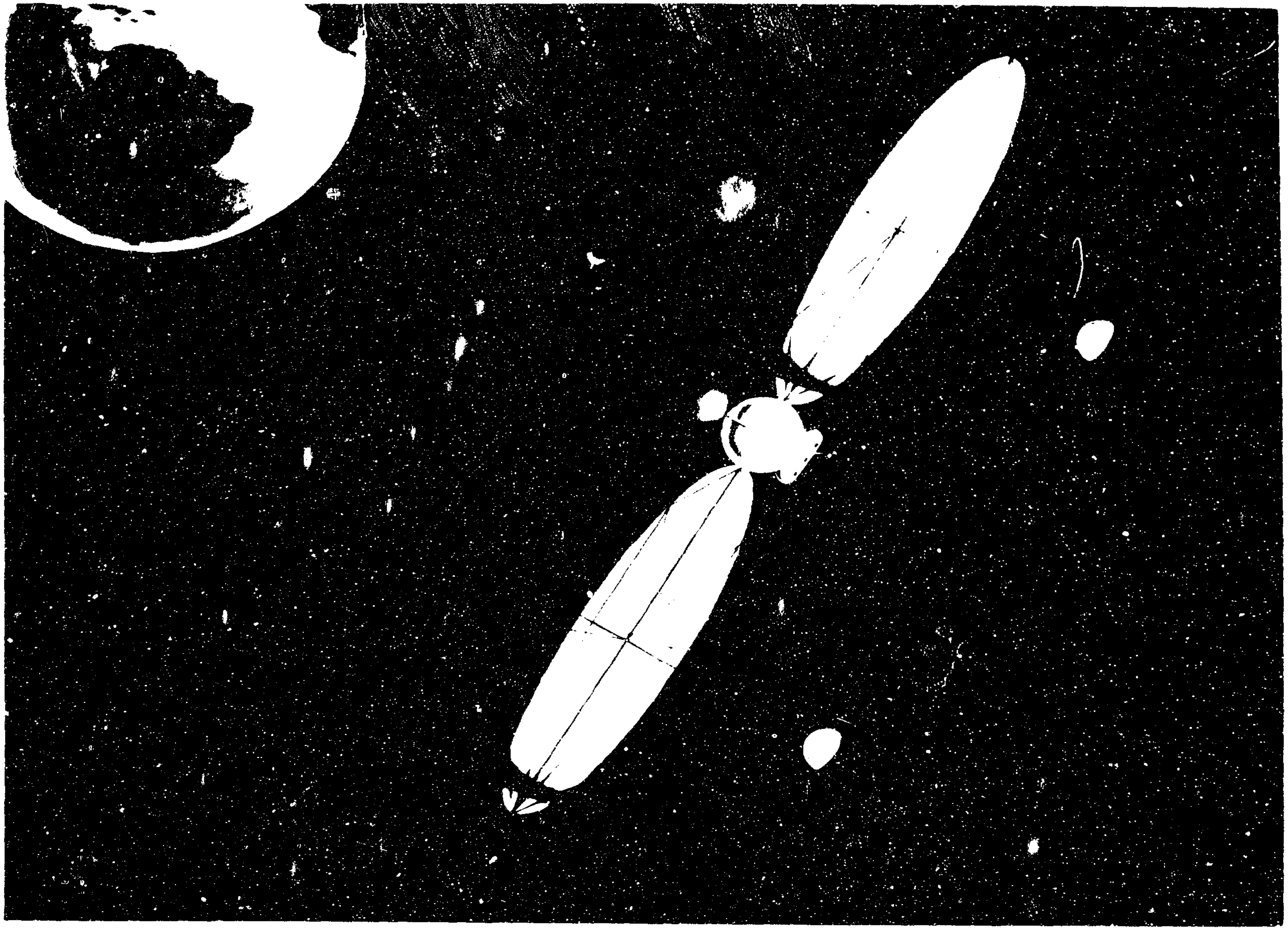

FIGURE 14. ENERGY CHARACTERISTICS OF SPHERICAL MIRROR (IN SPACE)


DETAIL B
1/2 SCALE
HEATING ELEMENT
HIGH PRESSURE PUMP RAISES LIQ. H2 PRESS TO 30 ATM
SOLAR BATTERY
LIQ. H2 TANK 11,000 LBS CAPACITY STORED AT 2 ATM PRESS
ROLL AXIS
SYMMETRY
A
SOLID THRUST MOTOR
.050 ALUMINUM LIQ. H2 TANK
DETAIL A
1/2 SCALE
SHUT OFF VALVES
GAS TANK FOR INITIAL PUMP OPERATION
ROLL CONTROL VANES 2 REQD (OPERATED BY SMALL ELECTRIC MOTORS)
FIBERGLASS
NOZZLE ACTUATING CYLINDERS ENERGIZED BY BLEED H2 FROM MAIN SUPPLY THRU SOLENOID VALVES
SINGLE STAGE IMPULSE TURBINE
2 STAGE CENTRIFUGAL PUMP SUPPLIES LOW PRESSURE LIQ. H2 TO HIGH PRESS PUMPS AT CENTER OF HEATING UNITS
COLD LIQ. H2 TUBE
INSULATION LAYER
TURNS WITH BUBBLE & TUBE
STATIONARY WITH H2 SPHERE
AXIS OF ROTATION ABOUT H2 SPHERE AND GONDOLA. (BUBBLES, MIRRORS AND CROSS TUBE ASSEMBLY ROTATE BY ELECTRIC DRIVE MECHANISM)
VIEW LOOKING IN DIRECTION OF FLIGHT
1/100 SCALE
PLAN VIEW
28 FT DIA SPHERE
SOLAR POWERED SPACE SHIP-PROTOTYPE




# SOLAR POWERED SPACE SHIP PROTOTYPE



The Solar Powered Space Ship